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ELECTRIC PROPULSION SYSTEMS FOR MARS MISSIONS

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ABSTRACT

The potential capabilities of electric propulsion systems for both unmanned and manned exploration of Mars are considered relative to those of other propulsion systems, primarily nuclear heat-transfer rockets. For unmanned explorations, a single electrically propelled vehicle, weighing about 25,000 lb and using a 250-450 kw power system with specific weights in the vicinity of 10 kg/kw, would be capable of performing most of the scientific interplanetary probe missions, not only to Mars, but to most planets and regions of the solar system. Chemical- or nuclear-rocket vehicles several times this initial weight can perform only part of these missions.

For manned exploration of Mars, the comparison depends on the specific powerplant weight attainable with electric propulsion and the specific impulse attainable with nuclear rockets. In general, if the specific weight is 10 kg/kw, electric-propelled vehicles require less initial weight, for a given mission, than nuclear rockets, but only at rather long mission times. As specific weight is reduced, the trip time for equal initial weight for nuclear and electric rockets moves toward lower values, reaching about 500 days and 400 days for specific weights of 6 and 4 kg/kw, respectively. Further reduction in specific weight makes electric propulsion superior at all trip times.

Consideration of shielding requirements for traversal of radiation belts and for crew protection during giant solar flares indicates that

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shielding equivalent to as much as 100 gm/cm^2 of graphite may be needed throughout the manned Mars mission. During the outward portion of the mission, propellant weight is more than adequate to provide this shielding, both for nuclear rockets and for electric propulsion. For the return trip, the best operation mode for electric-propelled vehicles consists of carrying reserve propellant equal to the shielding required for solar flares and using a reentry-landing vehicle to return the crew to Earth, thereby avoiding slow descent through the radiation belt and consumption of propellant during the last portion of the return trip. For nuclear-rocket vehicles, the best operation mode depends on the magnitude of solar-flare shielding required. For high values (about 100 gm/cm^2) or very low values, use of a reentry-landing vehicle, rather than a final Earth capture propulsive impulse, yields the lowest initial weight. When the shielding needed is about the same as the propellant weight required for the final Earth capture impulse, however, the best initial weights are obtained by using propellant to provide some of the capture impulse.

The relative initial weights required for nuclear- and electric-rocket propulsion were not appreciably altered by consideration of shielding requirements.

INTRODUCTION

Electric propulsion has been considered particularly suitable for planetary and interplanetary exploration since the earliest days of its conception. The relatively unrestricted specific impulse achievable appeared to guarantee reasonable propellant weight fractions for reaching even the more distant regions of the solar system in reasonable periods of time. Subsequent trajectory and mission studies confirmed these early expectations and defined the magnitudes of the important parameters (such as powerplant specific weight) required to produce attractive payload fractions for various missions. Preliminary system design studies indicated that these performance parameters should be attainable, and extensive research and development of suitable electric thrustors and power generation systems were initiated in the United States in 1957. Since that time, experimental work on electric thrustors, particularly ion thrustors, has been gratifyingly successful to the extent that there exists considerable confidence that thrustors with adequate efficiency and lifetime for interplanetary missions can be developed within the next few years.

The situation with regard to power generation systems for electric propulsion is less clear in that no system now under development can guarantee sufficiently low specific weight to be suitable for propulsion of interplanetary vehicles. The SNAP-8 system, which many people hoped could be used for Atlas-Centaur launched, electrically propelled vehicles, now appears to have little promise of weighing less than about 60 kg/kw as compared with the 30 kg/kw needed to make its use for interplanetary probes attractive. The SNAP-50 system, under development by the AEC and the Air Force, may be suitable for use with Saturn IB launched electrically propelled vehicles, but it is still too soon to determine whether the required specific weights and operating lifetimes will be achieved.

Despite the less-than-glowing near-term prospects for development of electrically propelled interplanetary vehicles, the long-range view is as attractive as always. There seems to be little doubt that, given enough research and development support, electric power generation systems with sufficiently low specific weight and long lifetime for at least the unmanned missions will be forthcoming. Current research on strong, high-temperature materials, with resistance to liquid-metal corrosion, should aid greatly in development of suitable nuclear-electric systems. In addition, there are some prospects of attaining lightweight power systems using thin-film techniques in solar photovoltaic cells (ref. 1) or with radioisotope direct conversion (ref. 2). Such methods eliminate high-temperature materials problems and constitute ultimately a more direct and simple approach to lightweight power systems. However, the extent of the success with these approaches cannot yet be clearly predicted.

In addition to the severe technical difficulties (of which other advanced propulsion systems also have their share), questions often arise concerning the desirability of or need for developing two or more advanced propulsion concepts and concerning the relative operational advantages of these systems. These questions have arisen particularly with respect to electric-rocket propulsion versus nuclear-rocket propulsion, since these systems are competitive for many future space missions, and both are now under development. As is usually the case, strong advantages and disadvantages can be pointed out for either system, and the discussions are likely to be more stimulating than conclusive. Nevertheless, I would like, in this paper, to address myself again to these rather nebulous questions. In particular, since this is a symposium on the exploration of Mars, I shall consider the questions of the relative suitability of electric propulsion and nuclear rockets for unmanned and manned Mars missions. This restriction to Mars mission imposes some penalty on electric propulsion, which tends to become more attractive as the distance to be traveled increases. However, if electric propulsion can be shown to have attractive competitive potential for Mars exploration, the increased attractiveness for more distant missions becomes self-evident.

UNMANNED EXPLORATION

As mentioned in the INTRODUCTION, the only electric power generation system now under development that may be suitable for propulsion of interplanetary vehicles is the AEC - Air Force SNAP-50 nuclear-turboelectric system. The goals for this system with regard to weight or electric power level are not yet completely firm, but levels in the 300-1000 kw range are mentioned, with specific weights hopefully in the range of 6-12 kg/kw. For the purpose of electric propulsion of interplanetary probes, the lower end of this range of power levels (as well as specific weights) appears of most interest. This is because the launching system most suitable for this range of power levels is the Saturn IB, which is to have the capability of placing about 28,000 lb into a near Earth orbit. Assuming that the electrically propelled vehicle should not exceed 28,000 lb and recognizing that the electric propulsion system should constitute about one-fourth of the vehicle weight for maximum payloads, we arrive at a maximum desirable power generation system weight of about 7000 lb. With the range of specific weights mentioned, this results in power levels in the range 250-450 kw. Higher power levels would not be useful for this purpose unless they were achieved without weight increase (i.e., at lower specific weight).

Shown in table I is a comparison of the payloads that could be carried by an electrically propelled space vehicle launched by a Saturn IB with those possible using larger nuclear-rocket and chemical-rocket vehicles. This comparison is based on calculations and data presented in reference 3. A total of 15 missions were compared in this manner in reference 3, but with a different electric-propelled vehicle. The conclusion to be drawn is much the same as in reference 3; namely, a single electrically propelled space vehicle, launched by what we can now call a "medium-sized" booster, can accomplish most of the unmanned exploration missions within our solar system. In contrast, much larger vehicles, using Saturn C-5, Nova, or nuclear boosters, can accomplish only a fraction of the desired missions if chemical and nuclear rockets are employed. Consequently, several space vehicles, of increasing size and increasing booster requirement, would have to be developed. Since the number of scientific interplanetary probe missions in the next decade or two may number several dozen, it appears that very large savings in development costs, booster costs, and launching costs would result from successful development of an electric-propulsion system in the size range indicated.

This is also the best reason for use of electric propulsion for unmanned Mars exploration. Obviously, such exploration can be accomplished with nuclear or even chemical rockets and, in fact, will to some extent be accomplished before an appropriate electrically propelled vehicle can be developed. More detailed surface exploration, however, with continuous television and data transmission may not be possible before the early or middle 1970's, at which time an electric-propulsion system of the required size may be available.

It is of some interest to estimate what could be done with power generator systems of the thin-film types mentioned in the INTRODUCTION, if they achieved successful development. For the solar photovoltaic type, electric power would vary during the mission as the inverse square power of the distance from the Sun; while, for the radioisotope systems, power decreases exponentially with time. Shown in figure 1 are relative payloads delivered to a low Mars orbit using solar power, radioisotope power, and constant nuclear-electric power, starting in each case with the same specific powerplant weight.* The curves indicate that the margin in specific weight needed by the variable-power systems to match the constant-power system is quite small. For missions closer to the Sun, of course, the solar-powered systems produce greater payload than the others, for a given initial specific weight.

If nuclear-turboelectric systems, for any reason, fail to achieve the specific weights of interest for electric propulsion, possibly these or other more direct conversion systems could be developed to fill the gap.

MANNED MARS EXPLORATION

Earlier studies of manned expeditions to explore Mars using electric propulsion (refs. 4-7) generally considered minimum-energy or direct trajectories both for outward and return trip. Because of the long waiting time at Mars required with such trajectories and the relatively long times needed to spiral out of, and back into, a low Earth orbit, these studies produced the impression that electric propulsion was inherently slow in comparison to chemical or nuclear rockets. Later work (ref. 8) showed

*These curves were obtained from trajectories calculated by Mr. John MacKay of the Lewis Research Center.

that, by following indirect return trajectories, similar to those used in studies of high-thrust missions, total round-trip times comparable with those with high-thrust systems were achievable if specific weights near 5 kg/kw were attained.

Since publication of reference 8, further advances in trajectory studies have been made to the extent that optimum trajectories and mission-time breakdown are available (refs. 9 and 10) for a wide range of total trip times and Mars residence times. Consequently, it is now possible to estimate with greater accuracy than before the weight requirements for manned Mars exploration expeditions as functions of the significant parameters. It is therefore desirable to revise previous comparisons (such as those of ref. 8) of nuclear and electric propulsion capabilities for this mission.

Shown in figure 2 is the simplest such comparison, which gives the payload fraction as function of total trip time, for several powerplant specific weights. The payload fraction is the ratio of the payload returned to a 300-km Earth orbit (or to the Earth's surface, in the cases of atmospheric braking) to the initial weight of the vehicle starting out from the same low Earth orbit. Included are all features of a complete round-trip mission (including descent to and escape from a 300-km orbit about Mars), except that disposal of supplies and waste products en route is neglected, and no weight is left at Mars. The electric-rocket curves were calculated from the trajectory data of reference 9, and nuclear-rocket curves by the method given in reference 8. The specific weights given are kilograms per kilowatt of actual jet power and therefore include the efficiency of conversion (assumed constant) of electric power into jet power. The trajectories given in reference 9 are of the constant-power, programmed-thrust type, which yield minimum propellant consumption but require large variations in specific impulse and thrust. Recently, round-trip trajectory data have been obtained by the method of reference 10 for the case of constant thrust and specific impulse. Spot comparisons indicate that there is little difference in predicted payload ratio between these two classes of optimized trajectories. Consequently, the weights calculated herein from the trajectory data of reference 9 can probably be closely attained also with optimum constant-thrust trajectories.

The comparison in figure 2 is considerably more favorable to electric propulsion than that obtained with the simplified, nonoptimum trajectories of reference 8. In particular, the curves for a given specific weight have shifted toward lower trip time by an amount corresponding almost to a doubling of specific weight; i.e., the curve in figure 2 for 4 kg/kw is about where the curve for 2 kg/kw fell in ref. 8 and similarly for other values of specific weight.

Of considerable interest in figure 2 is the effect of eliminating some of the propulsive requirements by relying on the Earth's atmosphere, instead of propulsion, to provide the final retarding impulse. For the case of the nuclear-rocket curves, the values using atmospheric braking were obtained by simply eliminating the last of the four major thrust periods required for an all-propulsive mission. Such results have previously been evaluated in reference 11 and approximately double the payload ratio for a given specific impulse. The effects of utilizing this maneuver with electric propulsion have not, apparently, been presented previously and are surprizingly large also. These values are, as yet, rather rough estimates. They were obtained from the data of reference 9 simply by eliminating the contribution to mass ratio due to the final descent to low Earth orbit and dividing by two the contribution of the heliocentric Mars-Earth transfer. The resulting increase in payload (or reduction in trip time), although not as sizable percentagewise as for the nuclear rocket, amounts nevertheless to the equivalent of about a 25-percent reduction in specific weight.

In general, the conclusions to be drawn from figure 2 are that, for the round-trip Mars mission, electric propulsion is superior to nuclear-rocket propulsion with regard to payload carrying capability if the overall specific powerplant weight is less than about 8 kg/kw and is superior in trip time, as well as payload, if the specific weight is reduced below about 4 kg/kw. The precise crossover points depend on the actual specific impulse attainable with nuclear rockets as well as on more detailed studies of actual weight disposal during the mission.

If the return payloads required for electric- and nuclear-rocket vehicles were identical, the curves of figure 1 would serve as a good indication of relative performance on a Mars mission. However, there are several factors that could make comparisons based on equal payload invalid. If, for example, the electric-propelled vehicle required heavy crew shielding because of relatively slow traversal of the radiation belts while the nuclear-rocket-propelled vehicle did not, or if the propellant used by one vehicle could serve all shielding requirements while that of the other could not, then obviously payload ratio in itself would have little meaning. Consequently, it is necessary to consider the possible shielding needs of the two systems.

ESTIMATES OF SHIELDING REQUIREMENTS

Numerous estimates have been made of the shielding required to protect crews from the variety of radiation sources encountered during space mission (see, e.g., refs. 12 to 18). The major sources of damaging radiation are: (1) the Earth's inner radiation belt (and possibly a Martian radiation belt, if it exists), (2) major and giant solar flares, (3) cosmic rays, and (4) the nuclear reactor of the propulsion system. All other sources (such as the more frequent minor solar flares and Earth's outer radiation belt) appear to be of negligible importance, in that the particles involved can be easily stopped by crew compartment walls of reasonable thickness.

The severity of the problem, in terms of shielding weight required, depends to a large extent on the radiation dose that will be considered permissible for the crew. At the present time, there is not sufficient evidence to permit firm establishment of allowable doses for various types of radiation. In reference 15, an acceptable emergency dose level, based on estimated recovery rate, is suggested which is somewhat more liberal than would currently be tolerated, but appears to be a good basis for minimum shielding weight estimates. The suggested total dose level is 300-600 rem for a period of time of the order of a 400-600 days, provided that no short-term dose greater than 50-80 rem is allowed. More conservative calculations would allow a total of 50 rem from long-term sources, such as cosmic rays, plus not more than 25 rem from each of the short-term sources (such as the Van Allen belt and solar flares). Future developments, such as medical means of reducing radiation damage or increasing recovery rates, may lead to larger allowable doses.

For total trip time less than 600 days, even the conservative allowance eliminates the need for cosmic-ray shielding, since the unshielded dose rate is about 0.66 rem/week. The nuclear reactor may be considered part of the propulsion system weight and will not be considered further in this paper, although the shielding provided for other sources will tend to reduce that required for the reactor.

Thus, the two major sources of radiation that may require extra shielding are the inner radiation belt and major solar flares. Sufficient shielding should be provided for each of these sources to limit the dose to the range of 25-80 rem depending on one's conservatism. Fortunately, both of these radiation sources are of relatively short duration, so that fairly confined quarters are tolerable for a "radiation storm shelter" design.

Inner Proton Radiation Belt

The duration of time in the inner Van Allen belt determines the dose received with a given shielding thickness. This duration in turn depends on the initial acceleration, or thrust-weight ratio, of the vehicle. For nuclear-rocket missions, the time is sufficiently short to impose no severe shielding requirements. A shielding thickness of about 5 gm/cm^2 is sufficient to limit the dose to less than 25 rem for the thrust-weight ratios typical of nuclear rockets (see ref. 11).

For electric-rocket propulsion, the time spent in the inner belt is shown in figure 3 as function of total trip time and initial acceleration. For each total trip time, there exists an optimum Earth escape time (t_1 in fig. 3) and, correspondingly, an optimum acceleration. For trip times less than 600 days, the minimum initial acceleration is about 2×10^{-4} g. The time t_2 in figure 3 is the time required to pass from a radius ratio of 1.20 to 2.00, which includes the most severe region of the Earth's inner radiation belt. For total trip times less than 600 days, this inner-belt traversal time is less than 10 days.

Fortunately also, only a fraction of this total traversal time need be spent in the belt itself. This is due to the confinement of the inner belt to a band spanning roughly $\pm30^{\circ}$ about Earth's magnetic equator. Shown in figure 4(a) is a sketch of the belt location; figure 4(b) contains a plot of the latitude variation of this belt relative to typical spiral escape trajectories.* The boundary of the belt is taken as the counting-rate contour of 10^3 protons/cm^2-sec, which corresponds to a level below which regular cabin-wall shielding provides adequate protection during the traversal period. Except during possible major solar flares, the crew can safety remain outside the shelter when the vehicle is not in the belt indicated.

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*These curves were prepared by W. Brunk of Lewis Research Center. They do not include the variation of radius of the trajectory with time. This variation affects the quantitative values of emergence and entry time, but has little qualitative effect.

The vehicle path, in terms of latitude as a function of time, is shown for three launch-plane inclinations (30°, 50°, and 90°) during typical series of revolutions. It is evident that the actual time spent in the belt depends strongly on inclination of the trajectory plane and is as much as 13 hours at a time for a 30° trajectory inclination. It may be as little as a few minutes during part of the traversal, with rather large periods outside. For higher inclinations, periods inside the belt are more evenly distributed and are mostly less than 1 hr. Consequently, a rather confined radiation shelter should impose no great hardship on the crew.*

Shown in figure 5 is an estimate, based on the calculations of reference 16, of the total dose as a function of graphite shielding thickness for several initial accelerations. Curves are shown for extreme cases of full time and 1/4 time actually spent in the belt during passage through the $\rho = 1.20$ to 2.00 radius zone. Figure 5 shows that, for a 500-day trip, corresponding to an initial acceleration of about 0.3 milli-g, a shielding thickness between 100 and 150 gm/cm^2 is required for a dose of 25 rems and between 50 and 110 gm/cm^2 for a dose of 80 rem. Taking intermediate values of immersion time and total dose, it appears that a value of 100 gm/cm^2 may be adequate for crew shielding during inner-belt traversal using electric propulsion.** Alternative procedures for reducing radiation-belt exposure, such as temporary thrust-level increases

*To achieve an orbital inclination of 50° rather 30° requires about 100 ft/sec greater launch velocity from Cape Canaveral, which would change the payload launching ability of a booster by less than 1 percent.

**No calculations are actually available for dose rate as function of shielding thickness for $\sigma > 100$ gm/cm^2, and extrapolation of results from references 16 and 18 differ quite significantly. Reference 18 yields higher apparent doses for $\sigma > 100$ gm/cm^2 and lower apparent doses for $\sigma < 100$ gm/cm^2. Results shown in figure 5 must therefore be regarded as very tentative until more detailed shielding calculations are available.

or letting the crew rendezvous with the main vehicle above the inner belt, are not considered herein.

Solar-Flare Shielding

Shielding required to protect against giant solar flares is perhaps even less well-established than that required for the radiation belt. Not only is the tolerable dose in question, but also the energy spectrum and flux of radiation from the most severe credible solar flare are not defined. Since relatively few of the giant flares have been observed, there is little statistical basis for severity estimates. Estimates of shielding requirements for a 25-rem exposure therefore range from some 160 gm/cm^2 of graphite or higher (ref. 12) down to less than 8 gm/cm^2 (ref. 14); the latter is based on a typical energy spectrum deduced in reference 17.

Shown in figure 6 is a comparison of estimated dose as function of shielding thickness obtained from various sources. It is evident that no definite conclusion regarding shielding required for solar-flare protection is possible at this time. If the lower curve is taken, the shielding problem for manned Mars missions is of minor importance. A more conservative viewpoint at this time is that shielding thicknesses equivalent to about 100 gm/cm^2 of graphite may be required for adequate giant-flare protection.

USE OF PROPELLANT FOR SHIELDING

If radiation shelter thicknesses of the order of 100 gm/cm^2 are required, it appears that as much use as possible should be made of the available propellant. For the outward trajectory through the Van Allen belts, enough propellant is probably available to provide more than adequate shielding with electric propulsion, as well as with nuclear-rocket propulsion. During the Earth-Mars transfer, also, sufficient propellant is available. During the return trip, however, the propellant weight can decline to zero with electric propulsion and could be absent entirely for the nuclear rocket, if atmospheric braking is relied upon to perform the Earth capture maneuver (as suggested in ref. 11). If large shielding weight must be maintained throughout the mission, however, it seems that a reserve propellant supply may be an excellent way to supply this shielding.

To estimate the minimum propellant weight needed as a function of shielding thickness, the geometry shown in figure 7 was considered, where the "storm shelter" is cylindrical in form and located at the center of the propellant tank. The shelter height h_O is taken as 7 feet. The shielding effectiveness of a material depends on its stopping power as well as density, as indicated by the equation

$$\rho(r_T - r_O) = K\sigma_O$$

where ρ is shielding material density, K is particle range in the material (relative to graphite), and σ_O is the graphite shielding density per unit area. The shielding thickness required is thus $r_T - r_O$.

Shown in figure 8 is the shield weight required for an eight-man crew as a function of effective graphite shielding for typical electric-rocket propellants (mercury and cesium) and nuclear-rocket propellant (hydrogen). In this calculation, shelter volume of 42 ft^3 per man was assumed, corresponding to a 7' × 3' × 2' space per man, which should be adequate for the moderately short-time periods of occupancy required. The relative ranges K are for protons of the hundred Mev energy range. It is seen that hydrogen is the most effective shield at low thicknesses, but that the high-density materials become superior for large shielding thicknesses. This is because, for large σ_0, the actual path length required for hydrogen, due to its low density, becomes very large, and thus makes the outer portion of the bulk rather ineffective relative to its weight. Figure 8 indicates that to provide 100 gm/cm^2 equivalent shielding requires about 100,000 lb of mercury or about 200,000 lb of hydrogen. These weights are considerably less than those available during the outward portion of a Mars mission but represent a considerably larger propellant weight than would otherwise be carried for reserve. However, in the case of electric rockets, mercury is as good as lead, in terms of weight, so that carrying this much excess mercury does not constitute a weight penalty if the shielding is required. For the nuclear rocket, a large portion of this hydrogen (possibly as much as one-half) could be used to provide the final Earth capture impulse, thereby eliminating the need for a fairly heavy reentry vehicle; or an additional shield of denser material could be provided for part of the shielding, thereby reducing the required hydrogen weight. More detailed mission study is required to determine the best operational mode for each propulsion system as a function of the terminal shielding weight needed. A preliminary version of such a study is reported in the next section.

Shown in figure 9 is the propellant weight required as a function of the number of men in the crew, assuming again 42-ft^3 shelter volume per man. The variation of required propellant weight with crew number is rather slow, particularly for hydrogen with heavy shielding.

Comparing the propellant weight needed for shielding with the propellant weight available during the Earth-Mars transfer portion of the journey indicates that the shelter volume, for $\sigma_0 = 100\ gm/cm^2$, could be larger than assumed. Thus, for a typical electric-rocket mission with an initial weight of 10^6 lb, approximately 600,000 lb of propellant are available at the start of the voyage, and about 250,000 lb remain upon arrival at Mars. Consequently, the shelter volume, using mercury propellant, could be about six times the 42 ft^3 per man assumed for the eight-man crew during the outbound trip. Alternatively, much better shielding protection (>175 gm/cm^3) could be provided during radiation-belt traversal, when such capability is particularly valuable.

For a typical nuclear-rocket mission with an initial weight of 1.8×10^6 lb, about 400,000 lb of propellant would be available after the Earth departure impulse. This weight, again, is adequate to provide a shelter volume several times the assumed 42 ft^3 per man size during the outward trip.

EFFECT OF SHIELDING REQUIREMENT ON INITIAL VEHICLE WEIGHT

To evaluate the effect of various assumptions regarding shielding requirements on relative initial weight, a particular mission was chosen. This mission is quite similar to those considered in references 8 and 11 and consists in sending an eight-man crew on an exploration mission, starting with a nuclear- or electric-propelled vehicle in a low orbit around Earth. An allowance of 10 lb/man/day is made for food, oxygen, water, and other consumable supplies: 50,000 lb are allowed for the (unshielded) crew cabin, including all environmental communications, navigation, and miscellaneous equipment; and 50,000 lb are allowed for a Mars landing craft capable of carrying part of the crew to the surface and launching them back to rendezvous with the mother ship, which remains in a low Mars orbit for a total of 25 days. In addition, when aerodynamic braking is used in place of the final Earth-capture propulsive impulse, an allowance of 30,000 lb for a reentry-landing vehicle is added to the return payload, as estimated in reference 11.

Results of the calculations of initial weight are presented for shielding requirements equivalent to 0 and 100 gm/cm^2 of graphite. Shown in figure 10 is the comparison for $\sigma_0 = 100\ gm/cm^2$. In this case, the best initial weights for the nuclear rocket, as well as for the electric rocket, are obtained if an atmospheric braking and landing vehicle is used. For the electric-propelled vehicle, the 100,000-lb shielding required (fig. 8) can consist of propellant reserve; while, for the nuclear rocket, a separate high-density shield of 100,000 lb is assumed. It may have been expected that better results could be achieved with the nuclear rocket by eliminating the 30,000-lb reentry vehicle and separate shielding and using hydrogen both for shielding and for the final Earth orbit capture impulse. However, the 200,000 lb of hydrogen needed for shielding during the Mars-Earth transfer (fig. 8) is much in excess of that needed for the final impulse and leads to higher initial weights.

Figure 11 shows the results obtained for $\sigma_0 = 0\ gm/cm^2$. For both nuclear and electric rockets, the best return mode in this case is the use of a reentry-landing vehicle in place of a final propulsive capture maneuver assuming that little or no reserve propellant is carried. This result, of course, is self-evident, in that, with no shielding or reserve requirement, it is best to use aerodynamic, rather than propulsive, capture maneuver.

Thus, both for high and low solar-flare shielding, atmospheric braking upon return to Earth leads to best initial weights, both with electric and nuclear rockets. For electric rockets, this method has the additional advantage of avoiding slow descent through the radiation belts when propellant supply is low. There is one condition for the nuclear rocket for which a propulsive capture is better than atmospheric braking. This is when the propellant needed for the final impulse is approximately equal to the shielding needed for solar-flare protection.

Comparison of figures 2, 10, and 11 shows that radiation shielding requirements do not appreciably change the relative performance capabilities of electric and nuclear rockets for the manned Mars mission.

CONCLUDING REMARKS

From the results of this brief study, it appears that consideration of particle radiation shielding problems should not appreciably affect the relative attractiveness of electric rockets and nuclear rockets for manned interplanetary exploration voyages. If electric-propulsion system specific weights of 4 to 6 kg/kw can be attained with adequate operational lifetime, electric propulsion shows definite superiority over nuclear rockets for manned Mars missions with regard to the initial weight required, even if specific impulses as high as 1000 seconds are ultimately achieved by nuclear rockets. For specific weights less than about 3 kg/kw, electric propulsion becomes superior with regard to total mission time as well as initial weight.

For unmanned interplanetary missions, electrically propelled vehicles tend to have greater versatility with regard to mission objectives and can achieve these missions with much smaller launching vehicles for equivalent payload. A specific weight of 10 kg/kw appears adequate to realize these advantages.

No attempt is made in this paper to assess the difficulty of achieving the desired performance and lifetime objectives relative to those of nuclear rockets. There is no basic reason why those objectives should not ultimately be achieved with both systems, difficult as they now seem. There is an obvious need for both of these propulsion systems if we wish to achieve the capability to explore at will and with reasonably sized vehicles not only the Moon and the near planets, but also the more distant reaches of our solar system.

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TABLE I. - PAYLOAD FOR SPACE PROBE MISSIONS

[From data of ref. 3]

Mission	Payload, lb			
	Electric propulsion 25,000 lb initial wt. 6250 lb powerplant wt.		Nuclear rocket 79,000 lb initial wt. "advanced" reactor	Chemical rocket 300,000 lb initial wt. H_2-O_2 propellant
	α = 6 kg/kw	12 kg/kw		
Mars Orbiter (500-mile orbit)	11,600 (250 days)	7600 (250 days)	18,500 (230 days)	30,000 (230 days)
Jupiter Flyby	12,200 (500 days)	8300 (500 days)	15,000 (500 days)	38,000 (700 days)
30° Out-of-Ecliptic	8,700 (232 days)	4400 (232 days)	4,000 (232 days)	No mission
Solar Probe (0.094 A.U.)	9,000 (200 days)	4600 (200 days)	No mission	No mission
Pluto Flyby	6,300 (1100 days)	2000 (1100 days)	No mission	No mission
Saturn Orbiter (2000-mile orbit)	5,200 (1000 days)	1100 (1000 days)	No mission	No mission
Jupiter Orbiter (2000-mile orbit)	4,000 (900 days)	300 (900 days)	No mission	No mission

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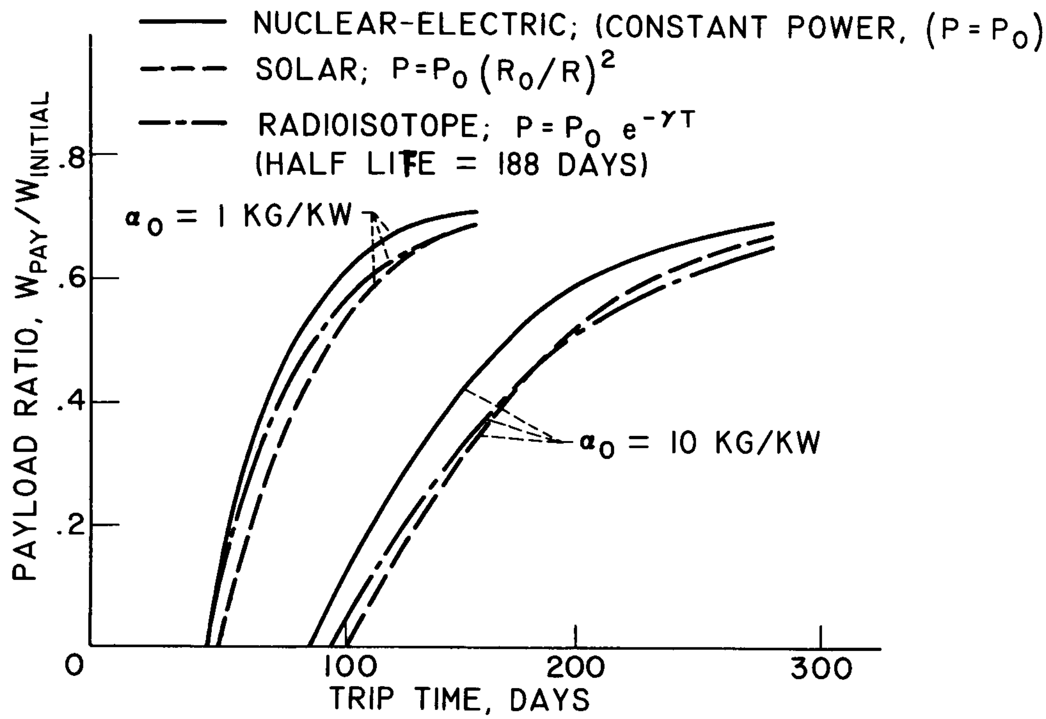

Fig. 1. - Mars probe payload for several types of power system.

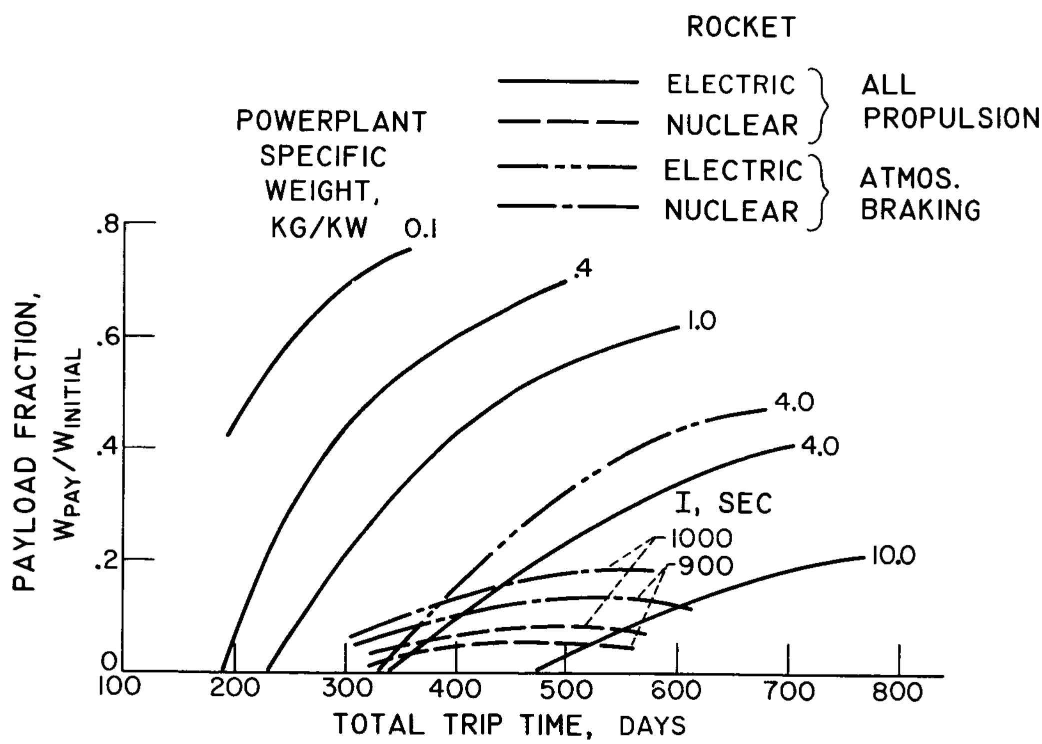

Fig. 2. - Payload for Mars round trip; 25 days in Mars low orbit.

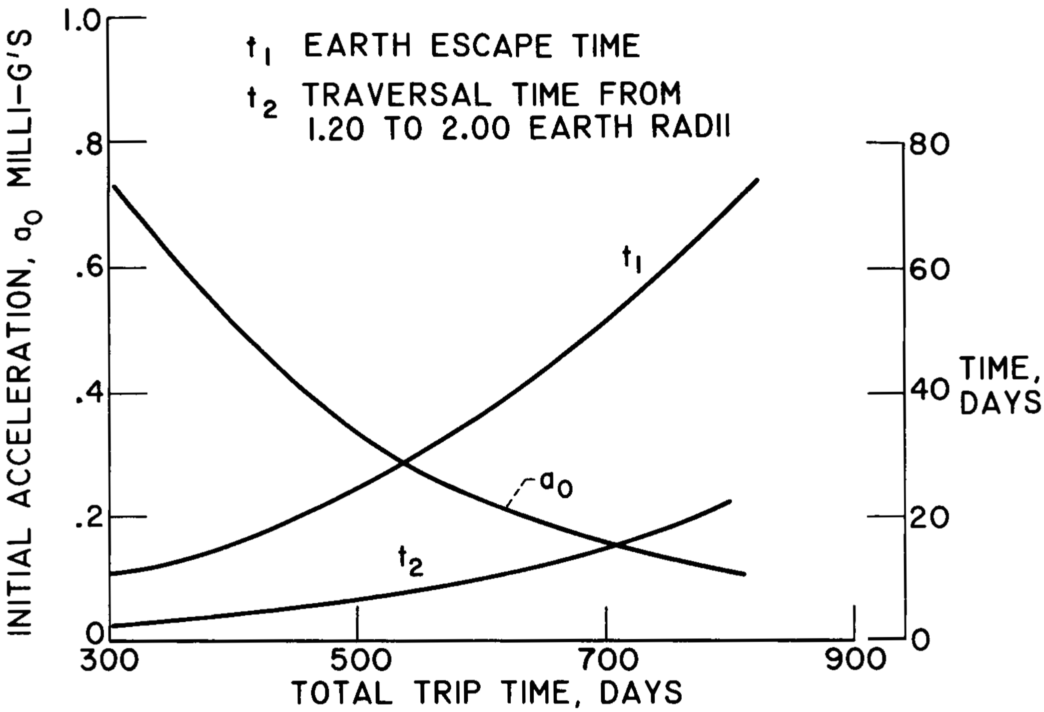

Fig. 3. - Initial acceleration and belt traversal time for Mars round trip.

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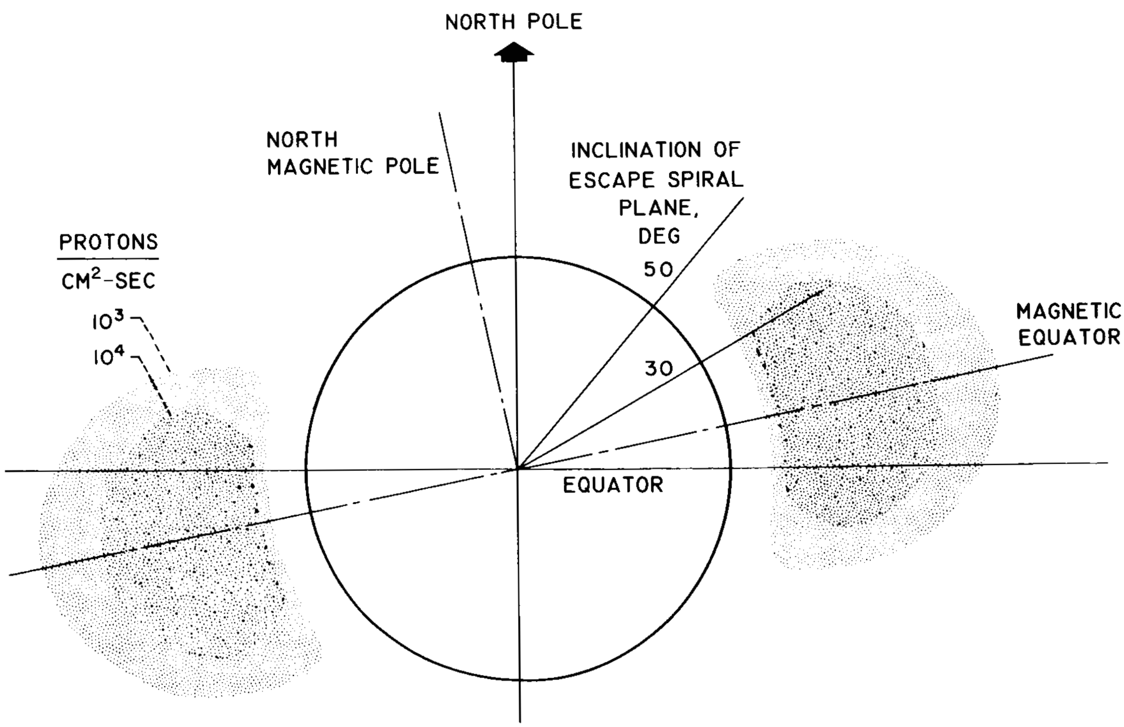

(a) Belt location relative to escape-spiral plane.

Fig. 4. - Low-thrust traversal of inner radiation belt.

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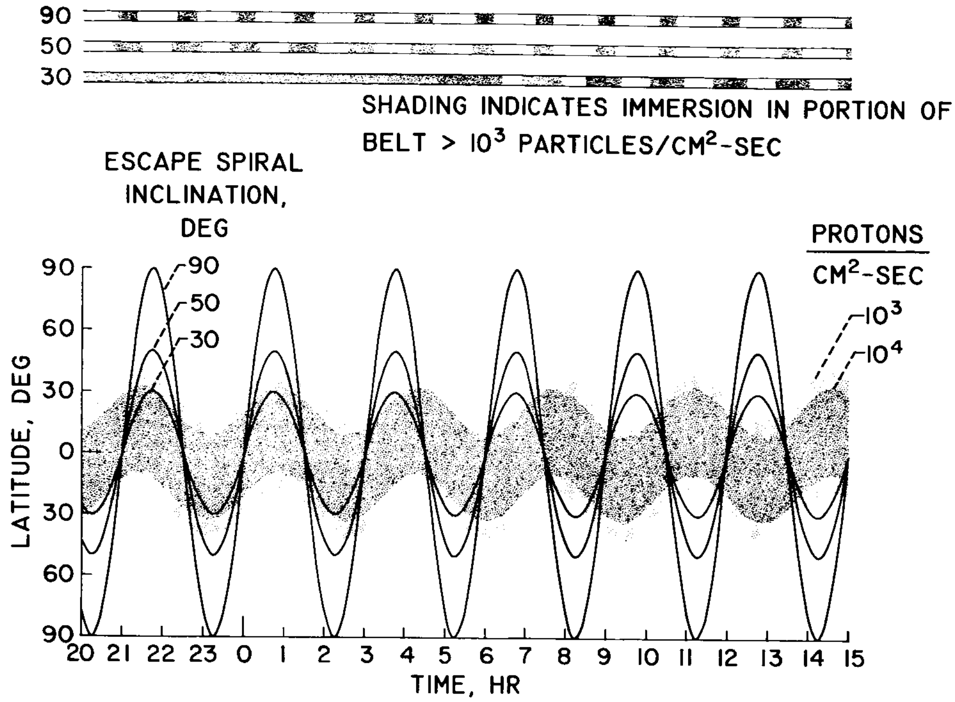


(b) Typical relation between escape trajectory and inner radiation belt. R = 1.7 R_E, 3-hr period.

Fig. 4. - Concluded. Low thrust traversal of inner radiation belt.

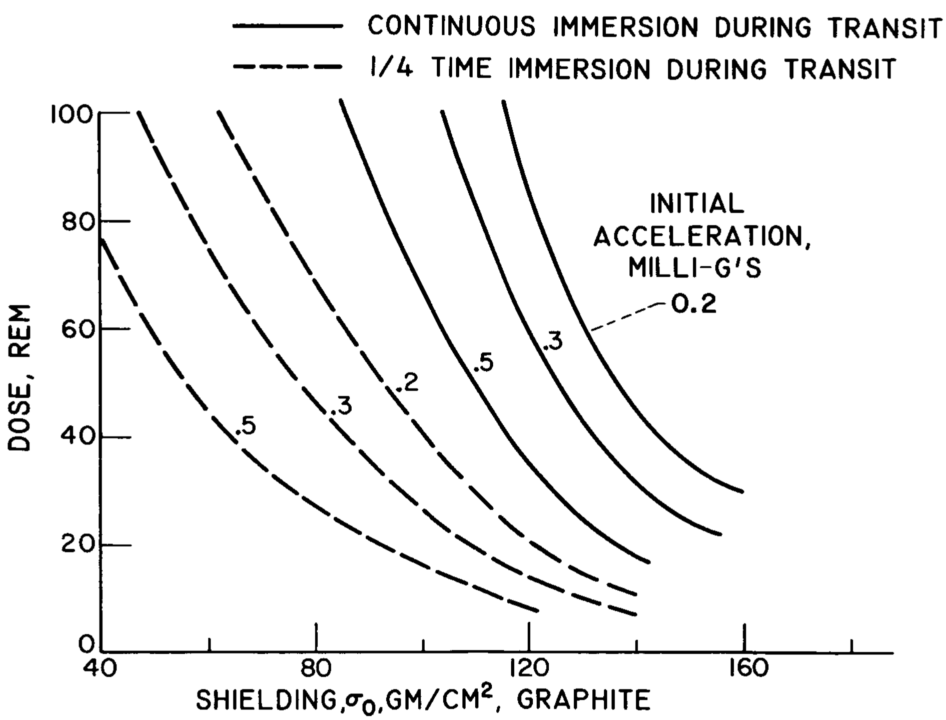


Fig. 5. - Shielding required for traversal of inner proton belt.

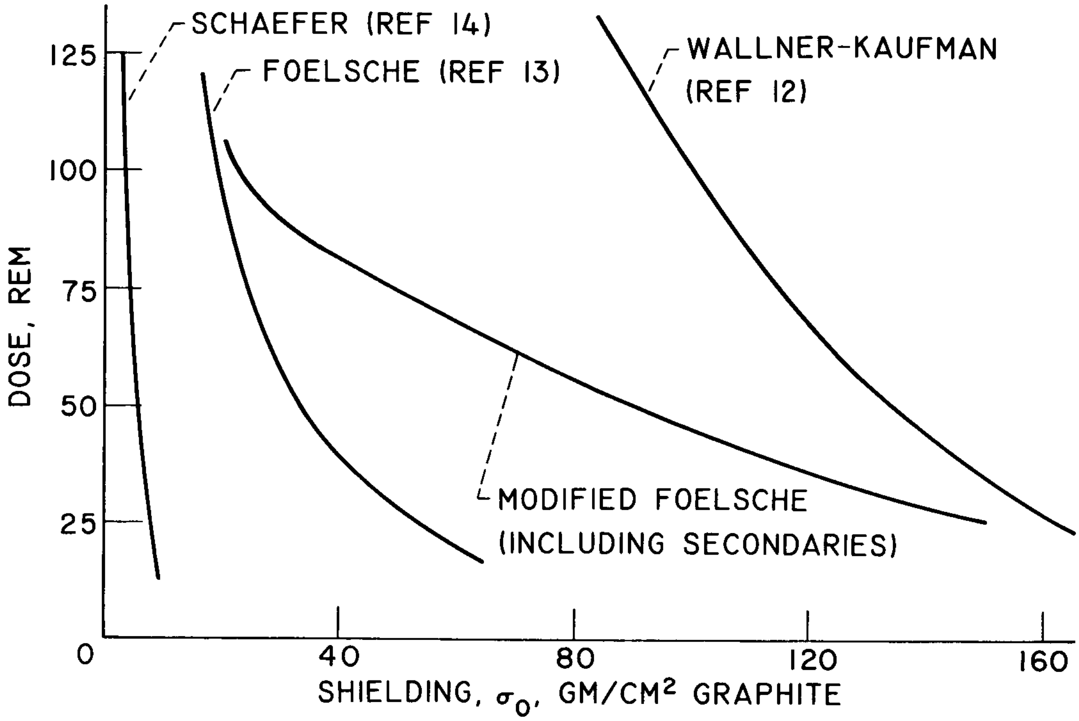
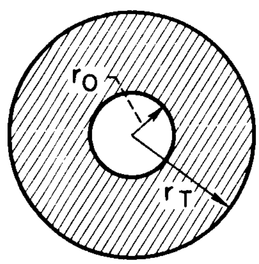
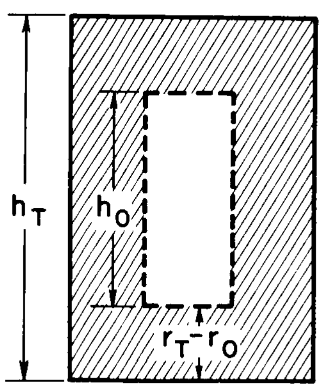


Fig. 6. - Shielding required for giant solar flares.



V_S = SHELTER VOLUME = $\pi\ r_O^2\ h_O$

V_T = TANK VOLUME = $\pi\ r_T^2\ h_T^2 - V_S$



h_O = 7 FEET

$\rho(r_T - r_O) = K\ \sigma_O$

ρ = SHIELDING MATERIAL DENSITY, LB/FT^3

σ_O = EFFECTIVE SHIELDING, LB/FT^2 GRAPHITE

K = PARTICLE RANGE, RELATIVE TO GRAPHITE

Fig. 7. - Shelter geometry considered.

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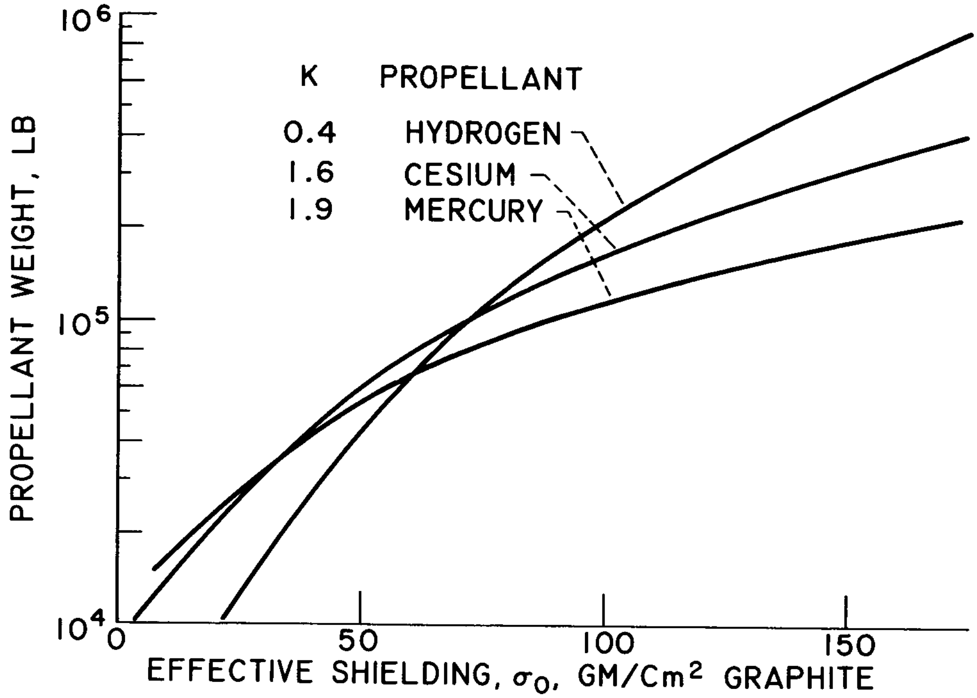

Fig. 8. - Propellant weight for radiation shelter. Cylindrical shelter; eight-man crew.

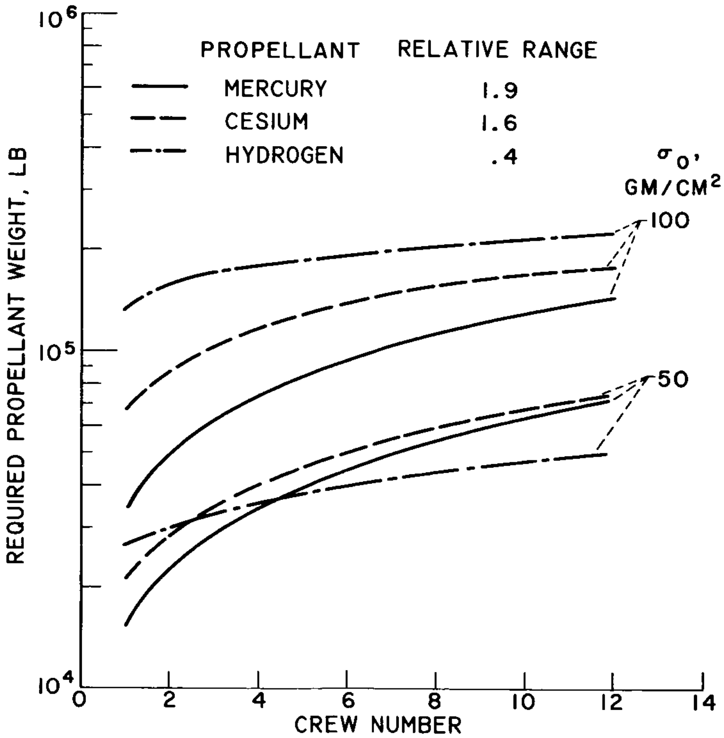

Figure 9. - Use of propellant for shielding. Cylidrical tank; 42 ft^3 per man.

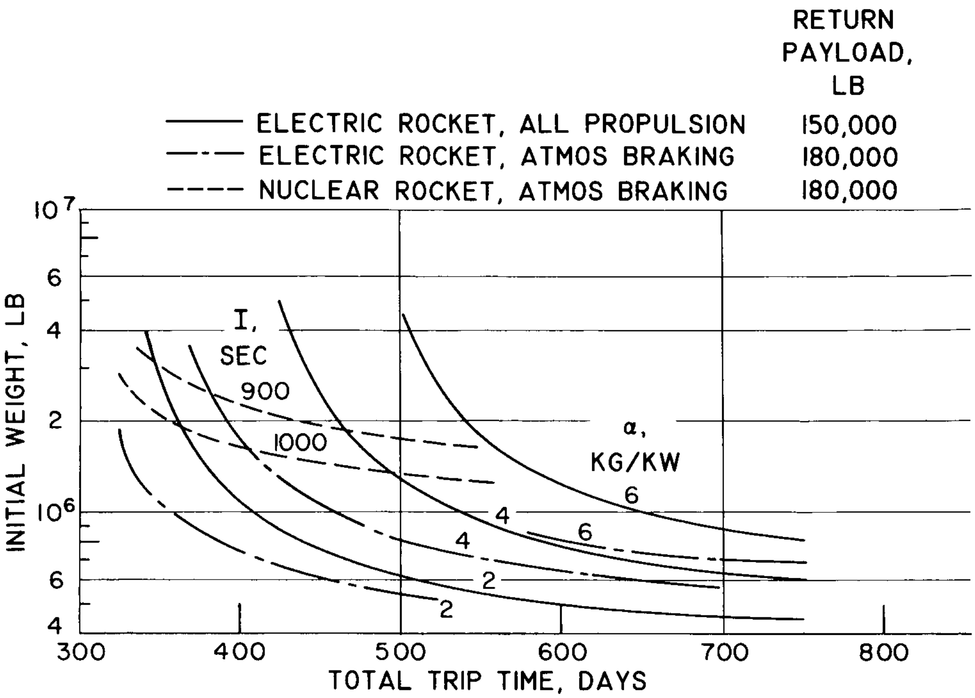

Fig. 10. - Initial weights for Mars round trip. 25 Days in low Mars orbit; $\sigma_0 \geq 100$ gm/cm^2 throughout mission.

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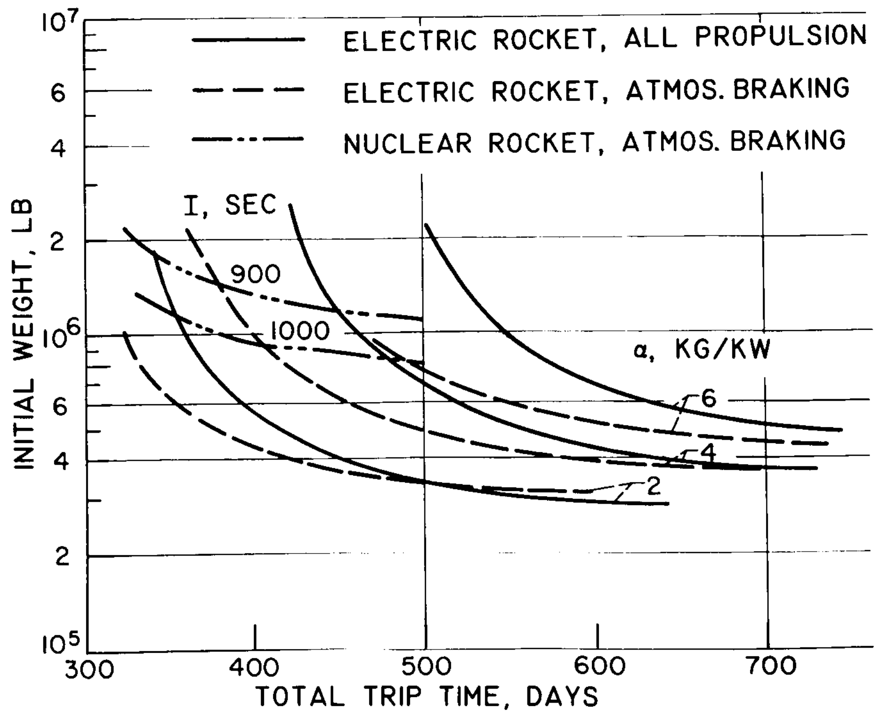

Fig. 11. - Initial weights for Mars round trip. Zero solar-flare shielding.